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DESIGN AND STRESS ANALYSIS OF SWEPT BACK WING WITH NASTRAN AND PATRAN

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Abstract - *A wing is a type of fin that produces lift, while moving through air or some other fluid. As such, wings have streamlined cross-sections that are subject to a3erodynamic forces and act as airfoils. A wing's aerodynamic efficiency is expressed as its lift-to-drag ratio. The lift generates at a given speed and angle of attack can be one to two orders of magnitude greater than the total drag on the wing. A high lift-to-drag ratio requires a significantly smaller thrust to propel the wings through the air at sufficient lift.*

In this article an attempt was made to bring out the challenges associated with the design of a Swept Back Wing for aircraft. A swept wing is a wing that angles either backward or, occasionally, forward, from its root rather than in a straight sideways direction. Wing sweep has the effect of delaying the shock waves and accompanying aerodynamic drag rise caused by fluid compressibility near the speed of sound, improving performance. The objective is to design and perform structural analysis of Swept Back Wing for aircraft with different load conditions and load factors. The Swept Back Wing is designed in CATIA tool and the linear static structural analysis is performed through MSC NASTRAN PATRAN software with satisfactory factor of safety.

Key Words: Swept Wing, CAD, CAE, FE Modelling

I. INTRODUCTION

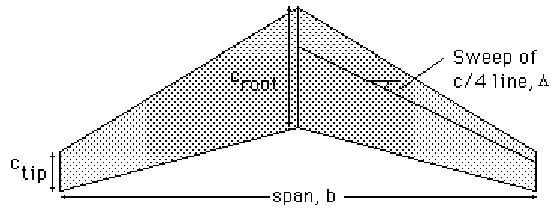
Wings develop the major portion of the lift of a heavier-than-air aircraft. Wing structures carry some of the heavier loads found in the aircraft structure. The particular design of a wing depends upon many factors, such as size, weight, speed, rate of climb, and use of the aircraft. The wing must be constructed so that it holds its aerodynamics shape under extreme stresses of compact maneuvers or wing loading.

Wing construction is similar in most modern aircraft. In its simplest form, the wing is a framework made up of spars and ribs and covered with the metal.
Spars are the main structural members of the wing. They extend from the fuselage to the tip of the wing. The entire load carried by the wing is taken up by the spars. The spars are designed to have great bending strength. Ribs give the wing section its shape, and they transmit the air load from the wing covering to the spars. Ribs extend from the leading edge to the trailing edge of the wing.

In addition to the main spars, some wings have a false spar to support the aileron and flaps. Most aircraft wings have a removable tip, which streamlines the outer end of the wing.

II. DESIGN CONSIDERATIONS

2.1 WING DESIGN PARAMETERS



S = wing area

$$AR = \text{aspect ratio} = \frac{b^2}{S} = \frac{b}{c}$$

$$\lambda = \text{taper ratio} = \frac{c_{tip}}{c_{root}}$$

Λ = sweep of c/4 line

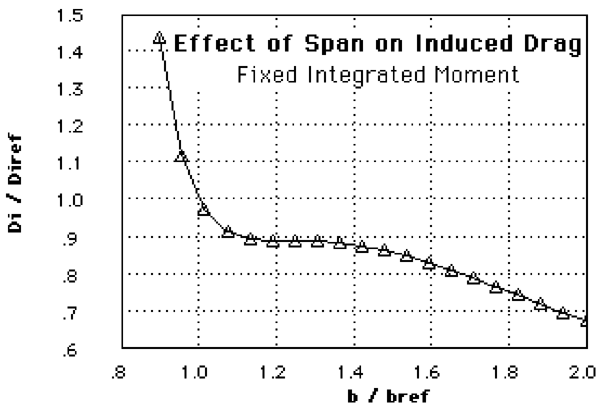
θ = wing twist angle

2.2 SELECTION OF WING SPAN

Selecting the wing span is one of the most basic decisions to made in the design of a wing. The span is sometimes constrained by contest rules, hangar size, or ground facilities but when it is not we might decide to use the largest span consistent with structural dynamic constraints (flutter). This would reduce the induced drag directly. However, as the span is increased, the wing structural weight also increases and at some point the weight increase offsets the induced drag savings. This point is rarely reached, though, for several reasons.

1. The optimum is quite flat and one must stretch the span a great deal to reach the actual optimum.
2. Concerns about wing bending as it affects stability and flutter mount as span is increased.
3. The cost of the wing itself increases as the structural weight increases. This must be included

so that we do not spend 10% more on the wing in order to save .001% in fuel consumption.
4. The volume of the wing in which fuel can be stored is reduced.
5. It is more difficult to locate the main landing gear at the root of the wing.
6. The Reynolds number of wing sections is reduced, increasing parasite drag and reducing maximum lift capability.



On the other hand, span sometimes has a much greater benefit than one might predict based on an analysis of cruise drag. When an aircraft is constrained by a second segment climb requirement, extra span may help a great deal as the induced drag can be 70-80% of the total drag. The selection of optimum wing span thus requires an analysis of much more than just cruise drag and structural weight. Once a reasonable choice has been made on the basis of all of these considerations, however, the sensitivities to changes in span can be assessed.

The wing span is chosen based on a wide variety of considerations including:

1. Cruise drag
2. Stalling speed / field length requirements
3. Wing structural weight
4. Fuel volume

These considerations often lead to a wing with the smallest area allowed by the constraints. But this is not always true; sometimes the wing area must be increased to obtain a reasonable C_L at the selected cruise conditions. Selecting cruise conditions is also an integral part of the wing design process. It should not be dictated a prior because the wing design parameters will be strongly affected by the selection and an appropriate selection cannot be made without knowing some of these parameters. But the wing designer does not have complete freedom to choose these, either. Cruise altitude affects the fuselage structural design and the engine performance as well as the aircraft aerodynamics. The best C_L for the wing is not the best for the aircraft as a whole. An example of this is seen by considering a fixed C_L, fixed Mach design. If we fly higher, the wing area must be increased by the wing drag is nearly constant. The fuselage drag decreases, though; so we can minimize drag by flying very high with very large wings. This is not feasible because of considerations such as engine performance.

2.2.1 EFFECTS OF SWEEP ANGLE

Wing sweep is chosen almost exclusively for its desirable effect on transonic wave drag. (Sometimes for other reasons such as a c.g. problem or to move winglets back for greater directional stability.)

1. It permits higher cruise Mach number, or greater thickness or C_L at a given Mach number without drag divergence.
2. It increases the additional loading at the tip and causes span wise boundary layer flow, exacerbating the problem of tip stall and either reducing C_{Lmax} or increasing the required taper ratio for good stall.
3. It increases the structural weight - both because of the increased tip loading, and because of the increased structural span.
4. It stabilizes the wing aero elastically but is destabilizing to the airplane.
5. Too much sweep makes it difficult to accommodate the main gear in the wing.

Much of the effect of sweep varies as the cosine of the sweep angle, making forward and aft-swept wings similar. There are important differences, though in other characteristics.

2.2.2 INFLUENCE OF THICKNESS TO CHORD RATIO

The distribution of thickness from wing root to tip is selected as follows:

1. It is likely to make the t/c as large as possible to reduce wing weight (thereby permitting larger span, for example).
2. Greater t/c tends to increase C_{Lmax} up to a point, depending on the high lift system, but gains above about 12% are small if there at all.
3. Greater t/c increases fuel volume and wing stiffness.
4. Increasing t/c increases drag slightly by increasing the velocities and the adversity of the pressure gradients.
5. The main trouble with thick airfoils at high speeds is the transonic drag rise which limits the speed and CL at which the airplane may fly efficiently.

2.2.3 Effects of taper ratio on Lift Coefficient

The wing taper ratio (or in general, the plan form shape) is determined from the following considerations:

1. The plan form shape should not give rise to an additional lift distribution that is so far from elliptical that the required twist for low cruise drag results in large off-design penalties.
2. The chord distribution should be such that with the cruise lift distribution, the distribution of lift coefficient is compatible with the section performance. Avoid high lift coefficients which may lead to buffet or drag rise or separation.
3. The chord distribution should produce an additional load distribution which is compatible with the high lift system and desired stalling characteristics.
4. Lower taper ratios lead to lower wing weight.
5. Lower taper ratios result in increased fuel volume.
6. The tip chord should not be too small as Reynolds number effects cause reduced C_l capability.
7. Larger root chords more easily accommodate landing gear.

Here, again, a diverse set of considerations are important. The major design goal is to keep the taper ratio as small as possible (to keep the wing weight down) without excessive C_l variation or unacceptable stalling characteristics. Since the lift distribution is nearly elliptical, the chord distribution should be nearly elliptical for uniform lift coefficients.

Reduced lift or t/c outboard would permit lower taper ratios. Evaluating the stalling characteristics is not so easy. In the low speed configuration we must know something about the high lift system: the flap type, span, and deflections. The flaps- retracted stalling characteristics are also important.

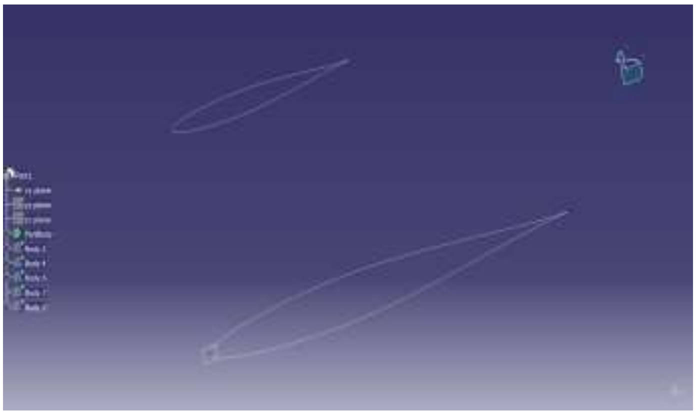
Design Parameters considered from trade-off studies	
Wing type	Low
Engine type	GE CF34-10A
Propulsion	2 turbofans
Cockpit crew	2
Passengers	100
Length(m)	33.875
Height(m)	8.79
Wing span(m)	27.3125
Wing area(m^2)	78.09982477
Wing sweep(deg)	25
Cruise(km/h)	800
Range(km)	3500
Ceiling(m)	12000
Thrust (KN)	69.2425
Stall speed(km/h)	229.3625
Fuel capacity(Kg)	10000
MTOW(Kg)	42000
OEW(Kg)	24500
Wing Loading(Kg/m^2)	537.7732937
V_{cruise} m/sec	222.2222222
V_{stall} m/sec	63.71180556

III. MODELING AND ANALYSIS

3.1 STEPS INVOLVED IN MODELING:

First to start with modeling of ribs we need to import the different airfoil coordinates of NACA 644-012 for root and tip of wing,

File >Import >Select file >ok



After importing the coordinates

Start >mechanical design >Wire frame and surfaces

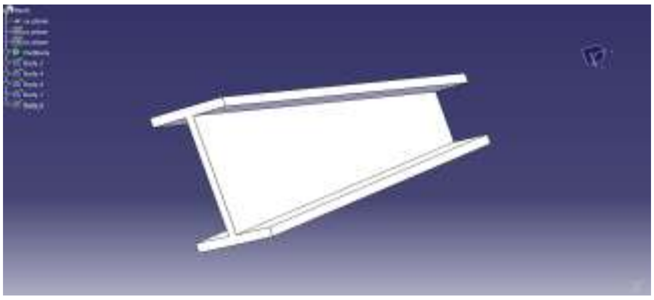
- To create the surface between the root and tip of the wing .

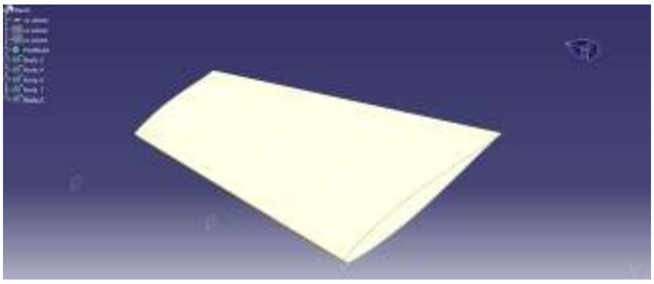
Surface menu >multi-section.

- To create the I section of the spar by using sketcher and sketch based features

Profile >Line >sketch based features >Pad

For the design consideration of spars the front spar located at the 30% of the chord length and the rear spar located at 70% of the chord length. The flange and web of the I section was drawn in particular dimensions by using the profile toolbar. And then using pad option in the sketch based features toolbar extrudes this section in the desired direction.

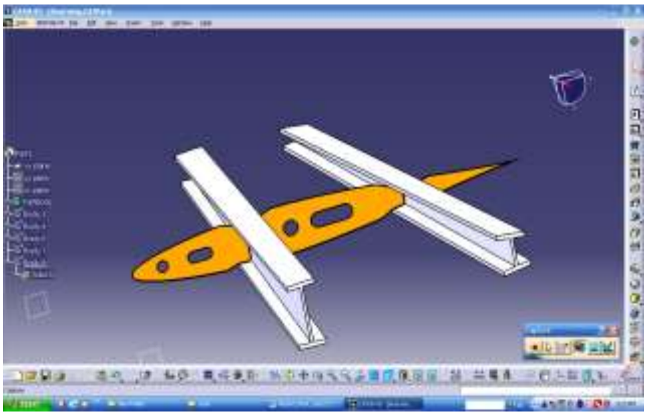




After designing the spars the ribs are designed by using the sketcher, split, pad, shell, pocket and fillet options. Now here we are using the split option because of the tapered wing design. Pocket option is used to create the weight reduction holes in the rib section. Fillet option is used to chamber the edge portion of the rib section in order to reduce the stress concentration.

To create ribs in the wing section by using the following steps:

Profile >Line > sketch based features >Pad >Split >Pocket >Dress up features >shell >fillet.



Phases of Finite Element Analysis

3.2 Pre-Processing

In this phase we will import CAD model for FE mesh by dividing the concerned subject geometry into sub domains for mathematical analysis as well as applies material features and boundary conditions.

3.3 Accessing, Importing & Exporting Geometry

In this phase the Designed CAD Model .CAT part file is imported into the PATRAN to perform the FEA analysis.

3.3.1 Geometry clean up

After importing the geometry into the PATRAN the geometry needs to be checked for any errors.

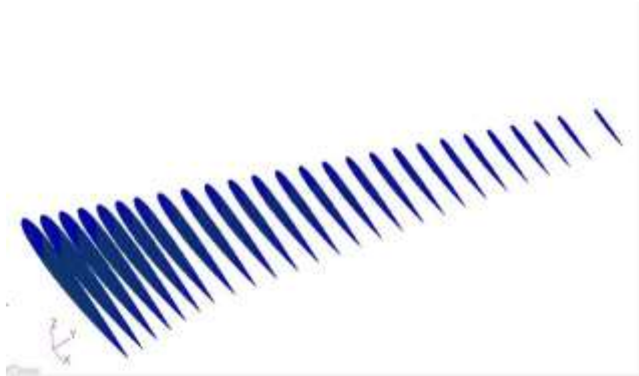
- Create a new database.
- Import the part file and update the model tolerance by applying the suggested value.
- Modify the geometry to be able to create a solid representation such as the boundary representation
- Verify the congruency in the geometry.
- Edit those surfaces that are no congruent.
- Create a trimmed surface using the defined outer loop and the reference surface generated during the disassemble step.
- Verify the congruency of the edited model and create a solid.

3.3.2 Grouping

After geometry cleanup the parts of the wing need to be grouped like the upper skin, lower skin, spars, and ribs. These commands are executed with the help of plot and erase command.

Upper skin

Lower skin



Ribs

Spars

3.4 DEFINING MATERIAL:

We define of what material our component is made with.
Create→Isotropic→Manual I/P→I/P properties→Elastic Modulus→Poisson Ratio→OK→Apply

3.4.1 DEFINING PROPERTIES:

The properties like Poisson ratio and Young's modulus corresponding to the material defined is given and application region is selected.
Create→2D→Shell→Prop.Name→Mat.prop.name→OK→Select member→Apply

3.4.2 Element properties:

The total skin is analyzed with 2D and 1D combination. Here the skin and spar web has assigned with PSHELL element and Spar flange is assigned with 1D Element.

3.4.3 DEFINING LOAD/ BC'S:

Since analysis of the rib and spar is to carried out which is under the load of the wing; hence loads are considered at the CG of the wing from where the loads are transfer to the ribs and then to the spars, we will define BCs at the nodes lying on the periphery of circular end of ribs.
Create→Force→Nodal→Set Name→Input Data→<F1 F2 F3>→Select application region→Apply

Create→Displacement→Nodal→Set Name→Input Data→ Translation<T1 T2 T3>→Select application region→Apply

3.4.4 DEFINING LOAD CASES:

In it, different cases of load/ BCs can be defined:
Create→ Load case name→ Assign Load/ BCs→ Select Individual Load/ BCs→OK→Apply

3.5 SOLVER CARRYING THE ANALYSIS:

Analysis is done in MSc/Patran by:
Analyze → Entire Model → Analysis deck → Translational Parameters → Select OP2 data output → Sub case Select→OK→ Apply
This is then submitted for solving to MSc Nastran by selecting required output file. The *.bdf file is generated. Processing is done using MSC Nastran; the command prompt is:
>>Nast 2004
>>filename.bdf

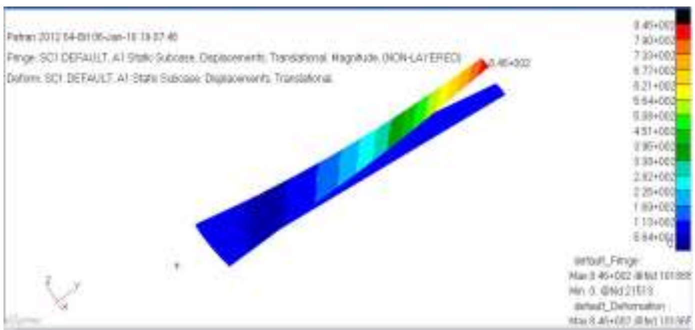
scr=yes old=no news = no

The *.f06 and *.op2 files are generated of which *.f06 file is checked for fatal error. MSC Nastran tool calculates global stiffness matrix, elemental forces from data given in loads and BCs. It then interprets the Matrices for geometry and give the displacement, stress and strain for model. Result can be verified by cross checking the reaction obtained from manual calculation and reaction in *.f06 file. Now *.op2 file is submitted to MSC/ Patran for post processing by:
Access Results→Read OP2→Result Entities→Select Result file→Apply

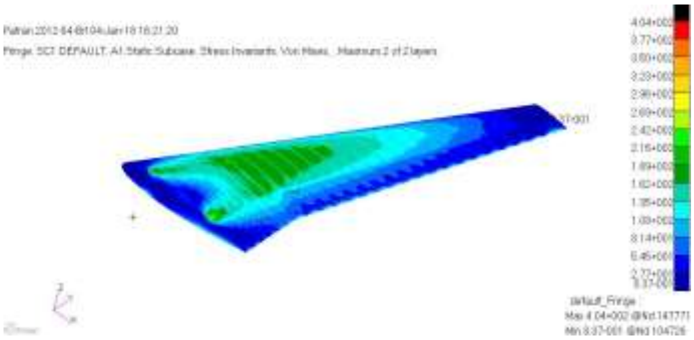
IV. RESULTS AND DISCUSSIONS:

A Various stress, force, displacement plots are available which can be accessed by clicking on Result icon and selecting Plot Markers in it.

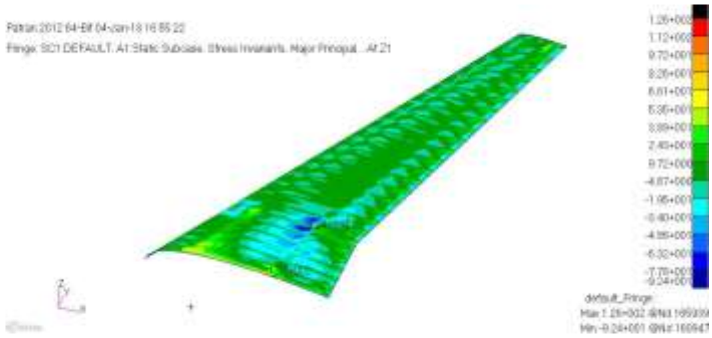
Case-I: Deflection at (2 mm skin thickness):



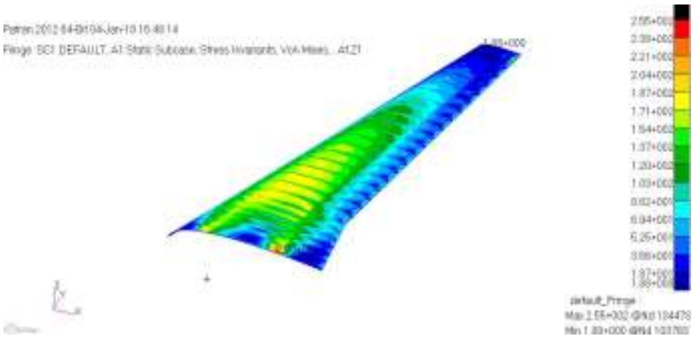
Deflection of the Wing



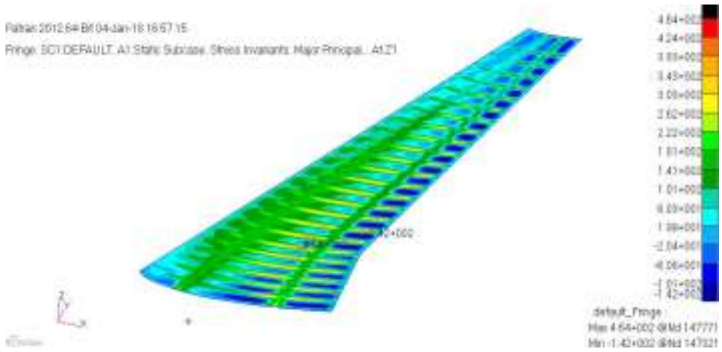
Von Mises stress of meshed wing structure



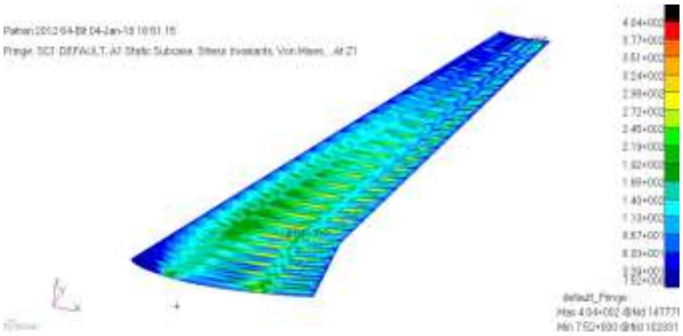
Maximum Principal Stress of meshed upper skin



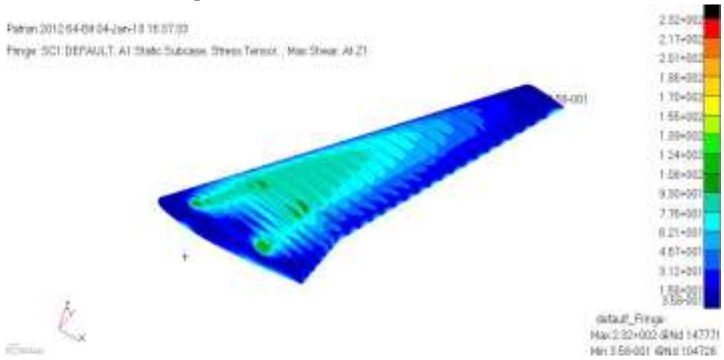
Von Mises stress of meshed upper skin



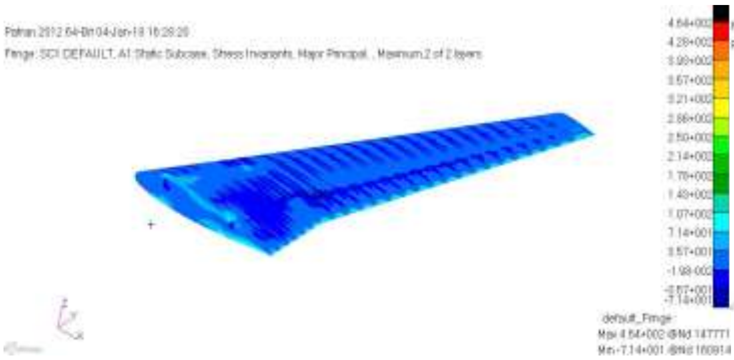
Maximum Principal Stress of meshed lower skin



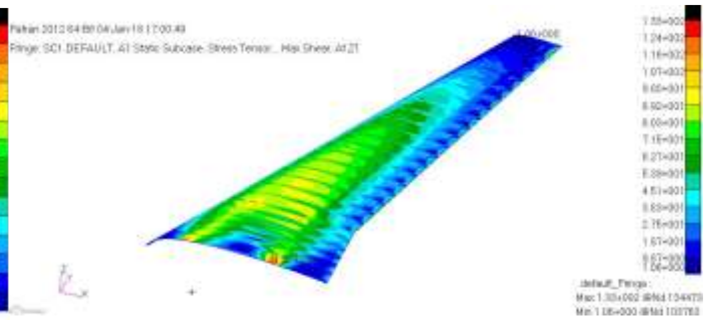
Von Mises stress of meshed lower skin



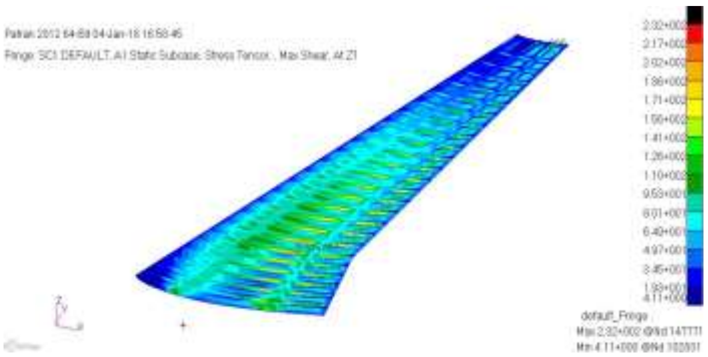
Maximum Shear stress of meshed wing structure



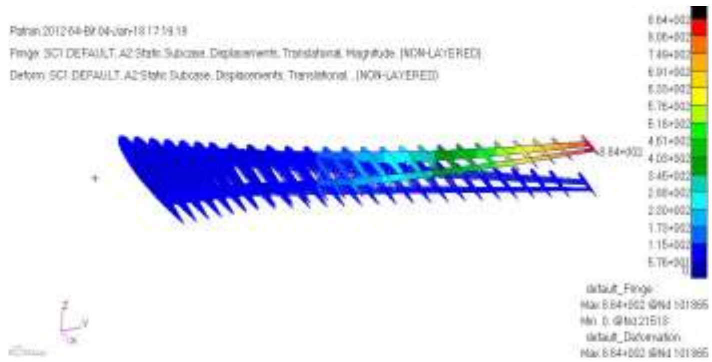
Maximum Principal Stress of meshed wing structure



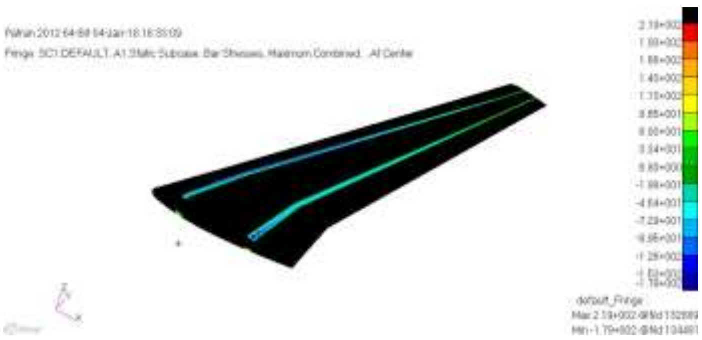
Maximum Shear stress of upper skin



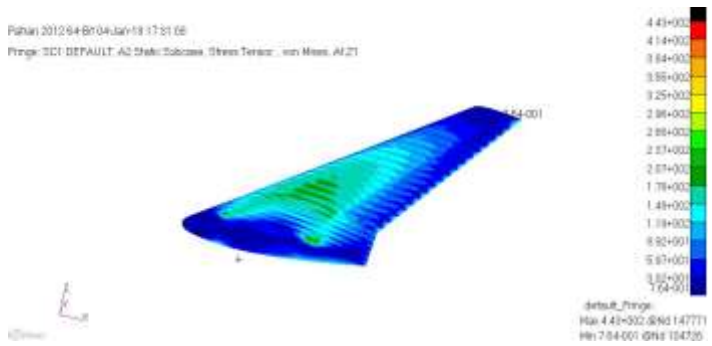
Maximum Shear stress of lower skin



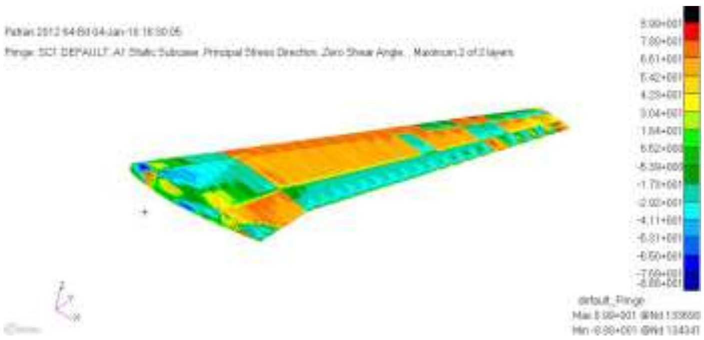
Deflection of meshed ribs & spars



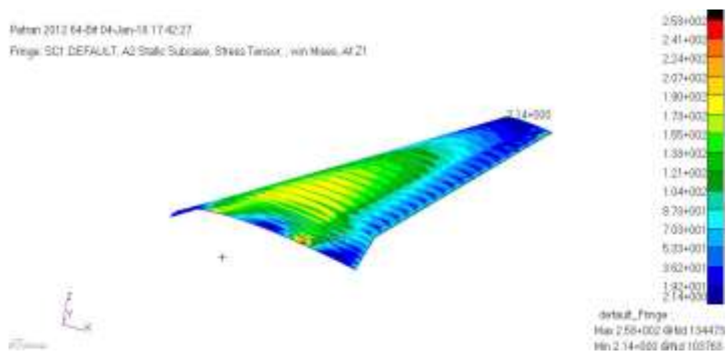
Maximum combined bar stress of a meshed wing structure



Von Mises stress of meshed wing

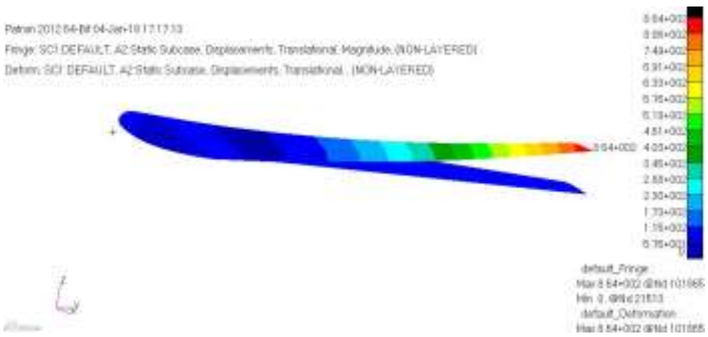


Maximum Principal Stress of meshed wing structure

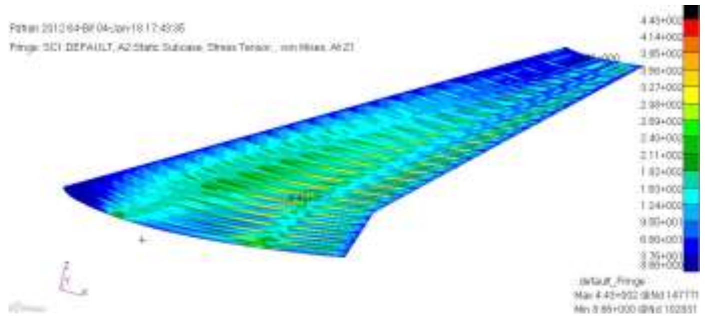


Von Mises stress of upper skin

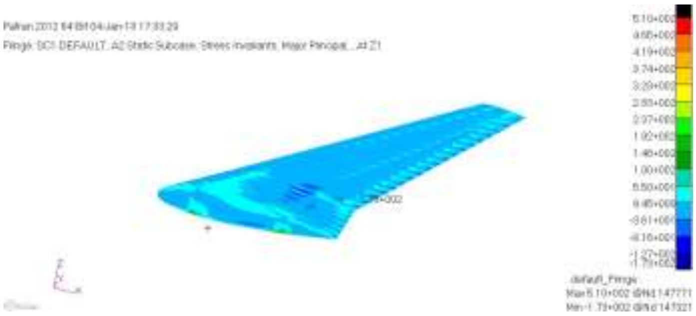
Case-II: Deflection at (1.8 mm skin thickness):



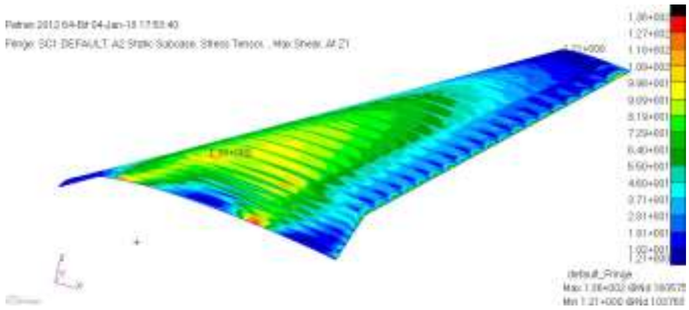
Deflection of meshed wing



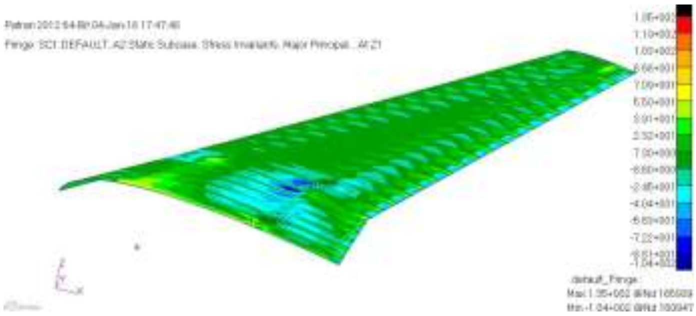
Von Mises stress of lower skin



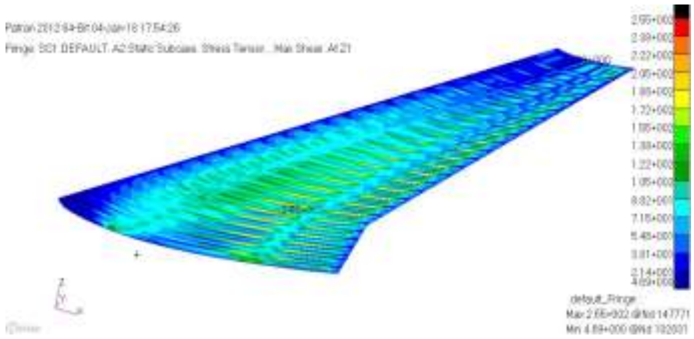
Major Principal Stress of meshed wing structure



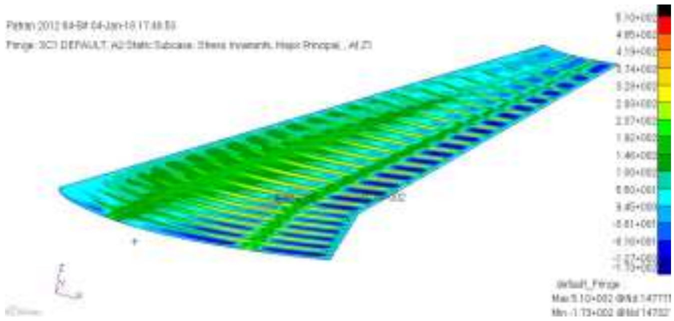
Max shear stress of meshed upper skin



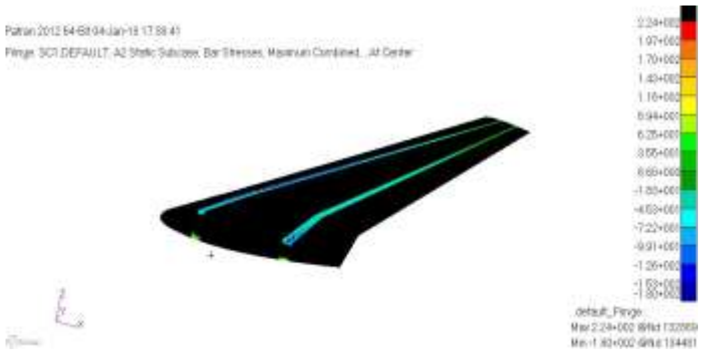
Major Principal Stress of meshed upper skin



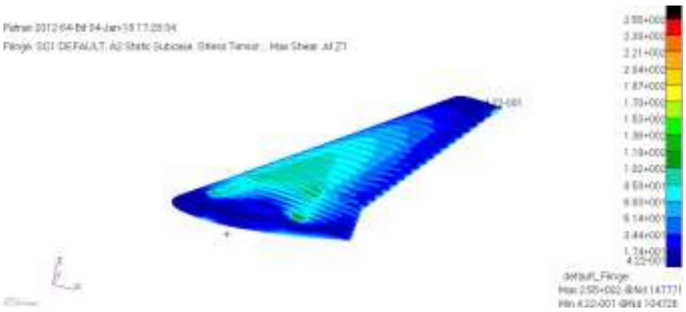
Max shear stress of meshed lower skin



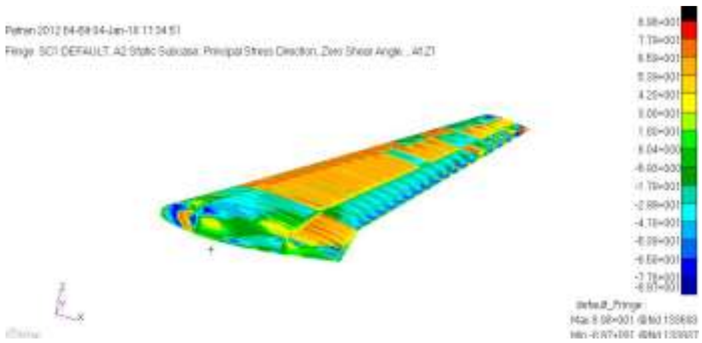
Major Principal Stress of meshed lower skin



Max combined bar stresses of meshed wing structure

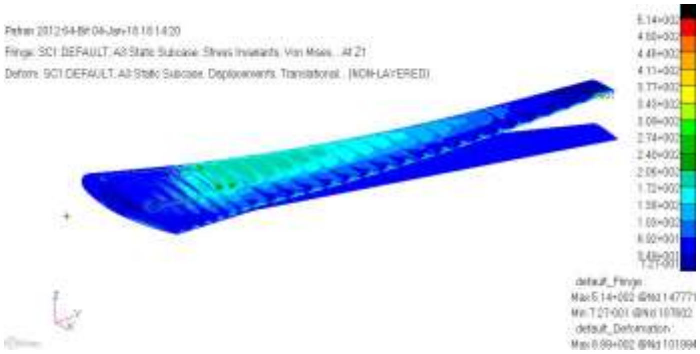


Max shear stress of meshed wing structure

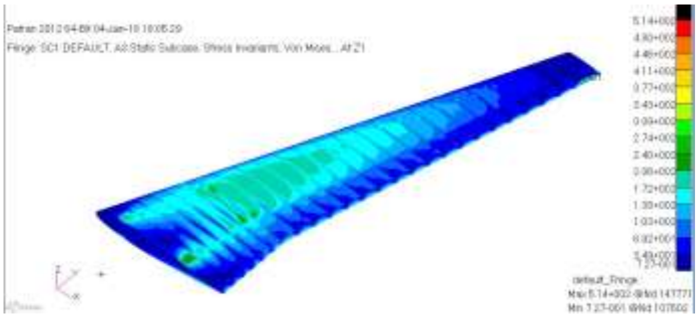


Max Principal Stress of meshed wing structure at zero shear angle

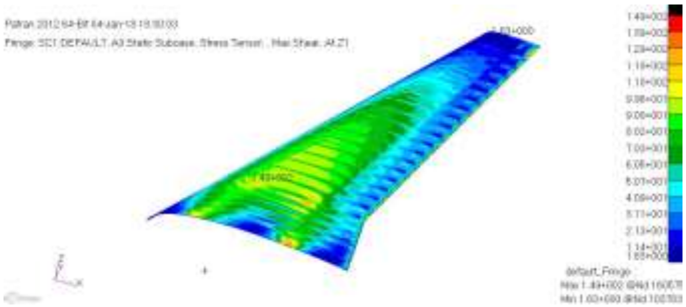
Case-III: Deflection at (1.5 mm skin thickness):



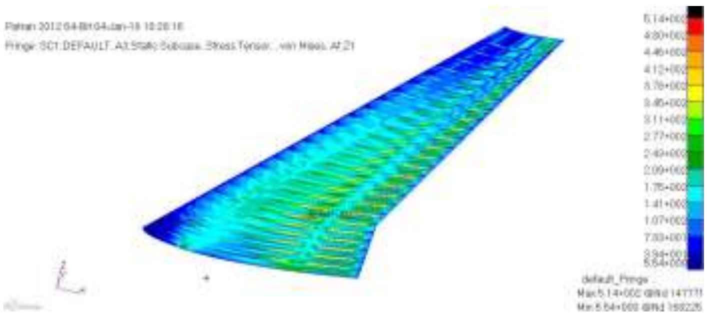
Deflection of meshed wing structure



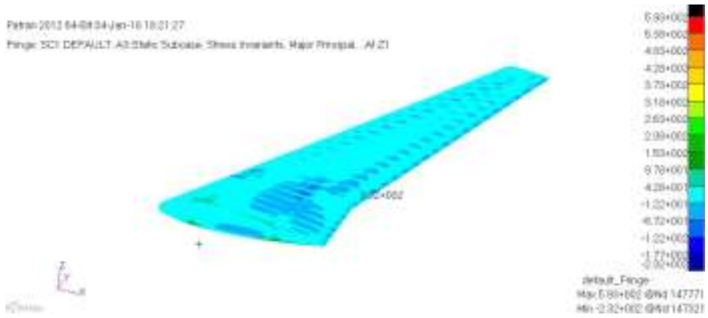
Von Mises stress of meshed wing structure



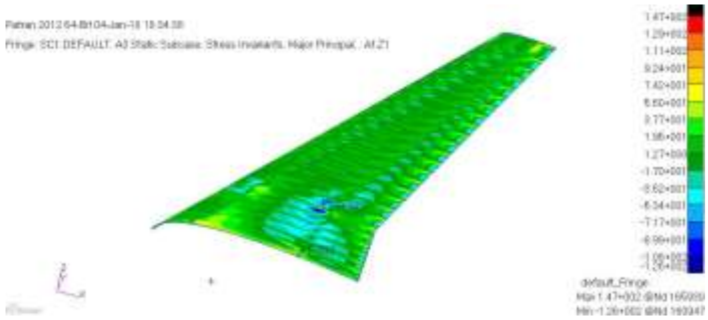
Von Mises stress of meshed upper skin



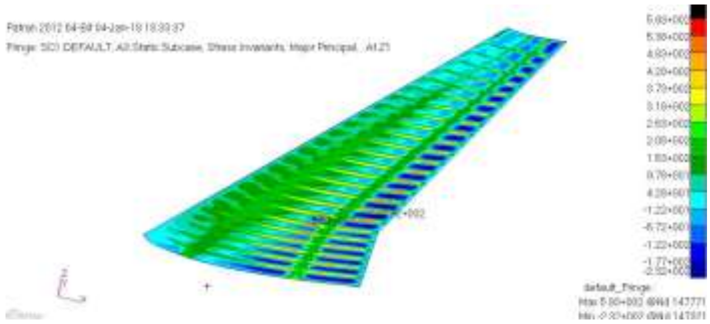
Von Mises stress of meshed lower skin



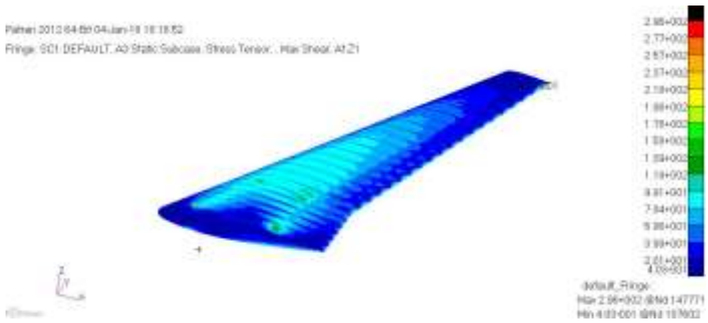
Major Principal Stress of meshed wing structure



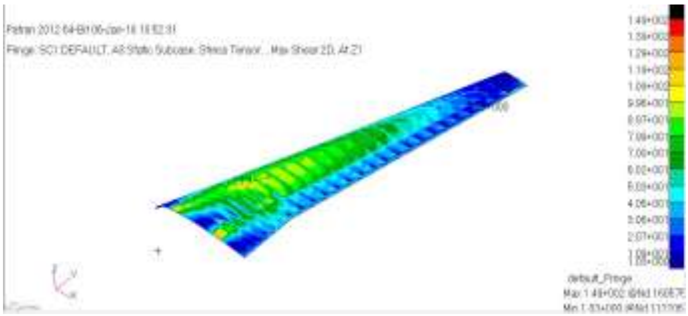
Major Principal Stress of meshed upper skin



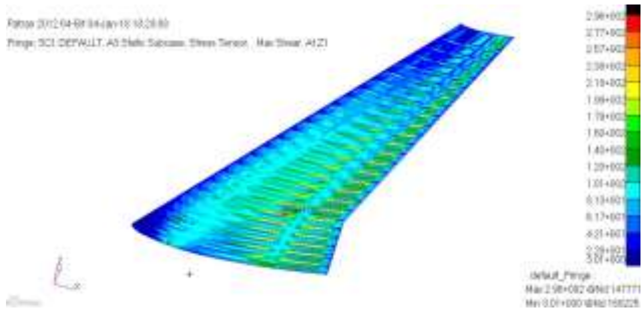
Major Principal Stress of meshed lower skin



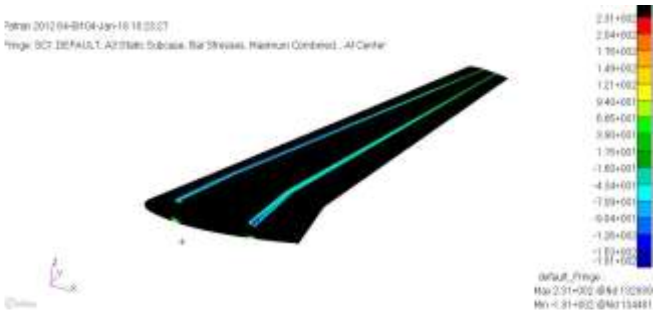
Max shear stress of meshed wing structure



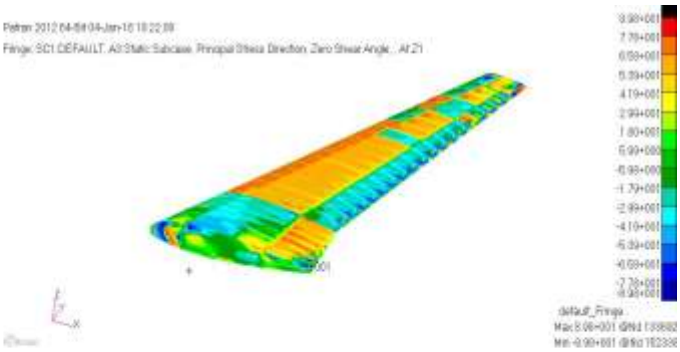
Max shear stress of meshed upper skin



Max shear stress of meshed lower skin



Max combined bar stresses of meshed wing structure



Maximum Principal Stress of meshed wing structure

V. DESIGN REVIEW

Displacements

Thickness	Deflection (mm)	Deflection in % of Span
2	846	6.9
1.8	864	7.12
1.5	899	7.41

1. Stresses for 2mm thickness

OVERALL WING STRUCTURE			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	404	583	1.44
Max Principle Stress	443	583	1.31
Max Principle Shear	514	349.5	1.134
UPPER SKIN			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	255	583	2.26
Max Principle Stress	126	583	4.63
Max Principle Shear	133	349.5	2.63
LOWER SKIN			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	404	583	1.44
Max Principle Stress	464	583	1.31
Max Principle	232	349.5	1.13

Shear			

2. Stresses for 1.8 mm thickness

OVERALL WING STRUCTURE			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	443	583	1.32
Max Principle Stress	510	583	1.46
Max Shear stress	255	349.5	1.37
UPPER SKIN			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	258	583	2.26
Max Principle Stress	135	583	4.32
Max Principle Shear	136	349.5	2.57
LOWER SKIN			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	443	583	1.32
Max Principle Stress	510	583	1.46
Max Principle Shear	255	349.5	1.37

3. Stresses for 1.5 mm thickness

OVERALL WING STRUCTURE			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	514	583	1.13
Max Principle Stress	593	583	0.98
Max Principle Shear	296	349.5	1.18
UPPER SKIN			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	149	583	3.91
Max Principle Stress	147	583	3.96
Max Principle Shear	149	349.5	2.38
LOWER SKIN			
VALIDATION	ACTUAL STRESS (Mpa)	ULTIMATE STRESS (Mpa)	R.F
Von-Mises	514	583	1.13
Max Principle Stress	593	583	0.98
Max Principle Shear	296	349.5	1.18

5.1 Validation of Reserve Factor:

Reserve Factor or Factor of Safety is defined as the ratio of Ultimate stress to the Actual stress or working stress.

$$RF = \frac{\sigma\ ultimate}{\sigma\ actual}$$

i. With Reserve Factor of greater than 1 where the structure can withstand the loads and is in factor of safety.
ii. With Reserve Factor of 2 or more is where the structure is in condition that can withstand loads beyond the working stresses.
iii. With a Reserve factor less than 1 where the condition of structure will fail for the working loads.

Here the above structure has Factor of safety greater than 1. So it means the Designed structure have good strength with all designed load cases. Then we can decrease structure or cross section dimensions of internal structure of wing. The advantages of decreasing the thickness or cross section will leads to minimal weight of the aircraft wing structure.

VI. CONCLUSIONS

Swept Back Wing design is analyzed based on the all design load conditions using NASTRAN and PATRAN. The swept back Wing is designed using (CATIA V5) software and the linear static structural analysis is performed through CAE (MSC NASTRAN PATRAN) software with satisfactory factor of safety and stiffness criteria's. It is observed that for 2 mm, 1.8 mm and 1.5mm thickness has better factor of safety.

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